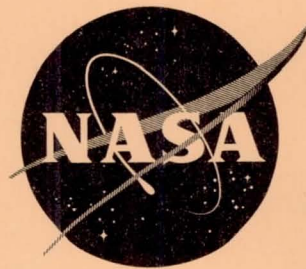


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TECHNICAL NOTE

D-858

DESIGN AND FLIGHT TESTS OF AN ADAPTIVE CONTROL SYSTEM
EMPLOYING NORMAL-ACCELERATION COMMAND

By Walter E. McNeill, John D. McLean,
Daniel M. Hegarty, and Donovan R. Heinle

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SUMMARY

An adaptive control system employing normal-acceleration command has been designed with the aid of an analog computer and has been flight tested. The design of the system was based on the concept of using a mathematical model in combination with a high gain and a limiter. The study was undertaken to investigate the application of a system of this type to the task of maintaining nearly constant dynamic longitudinal response of a piloted airplane over the flight envelope without relying on air data measurements for gain adjustment.

The range of flight conditions investigated was between Mach numbers of 0.36 and 1.15 and altitudes of 10,000 and 40,000 feet. The final adaptive system configuration was derived from analog computer tests, in which the physical airplane control system and much of the control circuitry were included in the loop. The method employed to generate the feedback signals resulted in a model whose characteristics varied somewhat with changes in flight condition.

Flight results showed that the system limited the variation in longitudinal natural frequency of the adaptive airplane to about half that of the basic airplane and that, for the subsonic cases, the damping ratio was maintained between 0.56 and 0.69. The system also automatically compensated for the transonic trim change.

Objectionable features of the system were an exaggerated sensitivity of pitch attitude to gust disturbances, abnormally large pitch attitude response for a given pilot input at low speeds, and an initial delay in normal-acceleration response to pilot control at all flight conditions. The adaptive system chatter of $\pm 0.05^\circ$ to $\pm 0.10^\circ$ of elevon at about 9 cycles per second (resulting in a maximum airplane normal-acceleration response of from ± 0.025 g to ± 0.035 g) was considered by the pilots to be mildly objectionable but tolerable.

INTRODUCTION

The use of stability-augmentation systems to provide satisfactory dynamic stability and response of high-performance airplanes over their full range of flight speeds and altitudes within the atmosphere has become standard practice. A common method of attaining a nearly constant desired response over the expected flight envelope has been to vary one or more system gains as appropriate functions of, for example, Mach number and altitude. Variation of these gains requires not only measurement of pertinent air data, but also fairly accurate prior knowledge of the variation of the basic airframe response with flight condition. Such changes are not always easily and accurately predictable.

The application of the concept of adaptive controls to the above problem as a means of obtaining a uniform optimum response over the flight envelope has offered attractive possibilities for some time, and various systems have been proposed (see, e.g., ref. 1).

The application of one type of adaptive control system, a normal-acceleration command autopilot using a limiter-type nonlinearity, to a high-speed air-to-air missile has been studied at the NASA Ames Research Center (ref. 2). The results of the study showed sufficient promise to justify a flight investigation of adaptive controls of a normal-acceleration type in a piloted airplane. A Convair F-102A interceptor was used for the tests.

The purpose of this report is to present and discuss the flight test results, which are in the form of normal-acceleration responses to elevator step inputs, and to compare them with results of a preliminary analog computer study. Pilot comments obtained during flight are also discussed. The major portion of the level-flight operational envelope of the test airplane was covered during the investigation.

NOTATION

A, a, B, b, C	arbitrary transfer function coefficients
A_N	normal acceleration (positive upward), ft/sec ² or g units
$A_{N,c}$	normal-acceleration command, g units or volts
C_L	lift coefficient, $\frac{\text{lift}}{qS}$
C_{L_α}	$\frac{\partial C_L}{\partial \alpha}$, per radian

$C_{L\delta}$	$\frac{\partial C_L}{\partial \delta}$, per radian
C_m	pitching-moment coefficient, $\frac{\text{pitching moment}}{qS\bar{c}}$
$C_{m\alpha}$	$\frac{\partial C_m}{\partial \alpha}$, per radian
$C_{m\dot{\alpha}}$	$\frac{\partial C_m}{\partial (\dot{\alpha}\bar{c}/2V)}$
$C_{m\delta}$	$\frac{\partial C_m}{\partial \delta}$, per radian
$C_{m\dot{\delta}}$	$\frac{\partial C_m}{\partial (\dot{\delta}\bar{c}/2V)}$
$\bar{c}$	mean aerodynamic chord, ft
G	a constant
g	acceleration due to gravity, 32.2 ft/sec ²
h	altitude, ft
I_y	pitch moment of inertia, slug-ft ²
K_1	forward-loop gain
K_2	steady-state aerodynamic gain $\frac{\dot{\theta}}{A_N}$, sec/ft
M	Mach number
N,P,Q	airplane $\frac{A_N}{\delta}$ transfer function coefficients
m	airplane mass, slugs
q	dynamic pressure, lb/sq ft
S	wing area, sq ft
s	Laplace operator, $\frac{d}{dt}$
t	time, sec

V	true airspeed, ft/sec
V_1	indicated airspeed, knots
x	unfiltered feedback signal, volts
x_1	feedback-signal component from pitch rate gyro channel, volts
y	input to limiter, volts
α	angle of attack, radians
$\dot{\alpha}$	$\frac{d\alpha}{dt}$
δ	elevator deflection, radians except as noted
$\delta_{a,c}$	aileron angle command, deg total deflection
δ_L	left elevon deflection, deg
δ_R	right elevon deflection, deg
ζ	damping ratio
ζ_0	damping ratio of second-order component of model transfer function
$\dot{\theta}$	pitching velocity, radians/sec
τ	time constant in first-order numerator of airplane transfer function, sec $\frac{\dot{\theta}}{\delta}$
τ_0, τ_1	time constants of first-order components of lead-lag networks in feedback loop, sec
τ_2	time constant of variable first-order term in denominator of model transfer function, sec
ω	frequency, radians/sec unless otherwise specified
ω_n	undamped natural frequency, radians/sec
ω_0	natural frequency of second-order component of model transfer function, radians/sec

EQUIPMENT AND INSTRUMENTATION

Airplane and Control System

A photograph of the test airplane is shown in figure 1.

The longitudinal flight control system in the test airplane was of the mechanical-hydraulic type with artificial pilot feel and added inputs from the MG-10 automatic flight control system (AFCS) through the pitch dampers. Inputs from the adaptive control system were added through the AFCS pilot-assist mode.

Adaptive Control System

The basic concepts governing the operation of the adaptive control system presently studied are described in detail in reference 2. Briefly, the system was a high-gain normal-acceleration feedback control system with a limiter installed in the forward loop to suppress the instability normally associated with high gain. System operation was characterized by a low-amplitude, moderate frequency limit cycle oscillation, or chatter, superimposed on the control system and airplane response. The limiter consisted of an operational amplifier with the normal feedback elements replaced by Zener diodes.

To aid in evaluating the performance of the adaptive system, a step-function generator was included which furnished a series of alternating positive and negative step normal-acceleration command signals up to a maximum of 2.5 g at intervals determined by the pilot.

To obtain pilot opinions of the airplane and system response to human commands, a standard B-4 flight controller (formation stick) was installed at the right side of the cockpit for pilot control while the adaptive system was operating. The command inputs introduced by the pilot were normal acceleration for pitch and aileron angle for roll. Electrical pilot trim adjustments in pitch and roll also were provided. The separate stick for pilot use during adaptive system operation was necessary because all elevon surface motions commanded by the AFCS, including the chatter, were transmitted back to the center stick. Further, any attempt to move the center stick an amount corresponding to more than about 1° of elevon would cause the AFCS to disengage.

Recording Instrumentation

The pertinent measured flight quantities were recorded by means of an 18-channel photographic oscillograph.

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ADAPTIVE SYSTEM DESIGN AND GROUND SIMULATION

Range of Simulated Flight Conditions

In order to assess the applicability of the proposed adaptive system to best advantage, as large a range of flight conditions as practicable was selected for preliminary study and for later flight tests. The five combinations of altitude and Mach number or airspeed chosen are as follows:

Condi- tion	V_i , knots	M	h, ft
1	200	0.36	10,000
2	---	.94	12,000
3	---	1.15	30,000
4	---	.80	40,000
5	---	.95	40,000

A list of the pertinent aerodynamic, inertial, and dimensional parameters of the test airplane is presented in table I and the airplane A_N/s transfer function coefficients are presented in table II.

Basic System

A simplified block diagram of the system as envisioned at the beginning of the present study is shown in figure 2. The "model" concept of operation was used. The theory of operation explained in reference 2, is that a sufficiently high forward-loop gain K_1 causes the closed-loop transfer function $A_N/A_{N,c}$ to approximate the model transfer function, which is the reciprocal of the feedback transfer function.

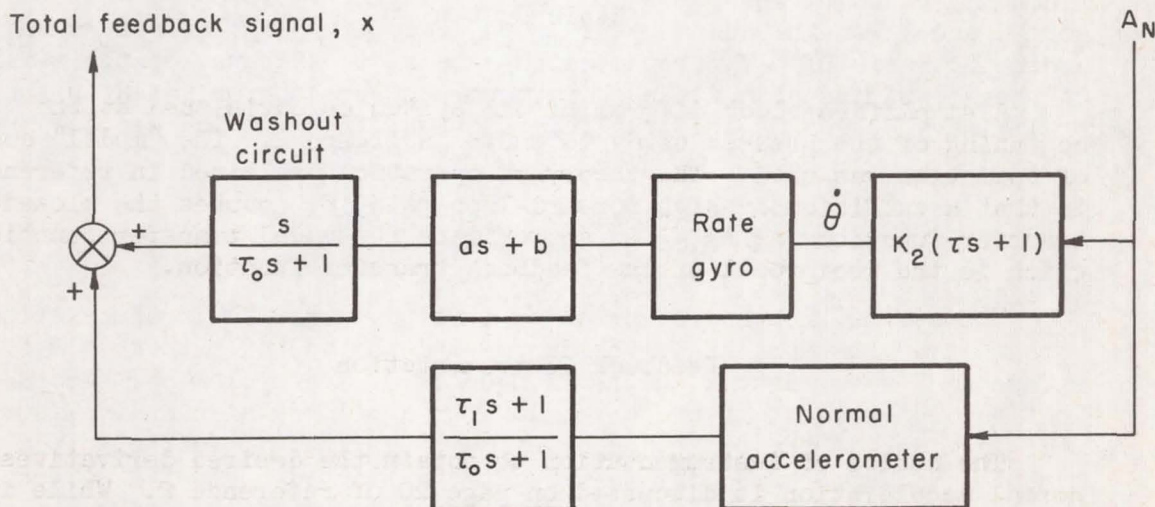
Feedback Instrumentation

The choice of instrumentation to obtain the desired derivatives of normal acceleration is discussed on page 20 of reference 2. While it would be desirable to obtain the entire feedback information using the normal accelerometer installed as part of the AFCS, it is well known that the noise in the output of the usual normal accelerometer is considerably higher than that in the output of a rate gyro. For this reason, the method used to obtain the derivatives of normal acceleration for use in the present system (fig. 2) was to take advantage of the airframe dynamic relationship

$$\frac{\dot{\theta}}{A_N} \approx K_2(\tau s + 1)$$

(ref. 3, pp. 19 and 21). This was accomplished by combining pitch rate and its first derivative with normal acceleration. The method of synthesis of the desired feedback signal is described in appendix A. A more detailed analysis of this approach is given in reference 2. Generation of the feedback signal in this manner for the present adaptive system, however, meant that the A, B, and C in figure 2 and hence the model must change with flight condition.

At the low flight speeds of the F-102A (as compared with the missile of ref. 2), the value of K_2 was sufficiently large to create changes in C which could seriously alter the steady-state gain $A_N/A_{N,c}$ of the adaptive airplane (see appendix A). Since it was desired to maintain a constant steady-state value of $A_N/A_{N,c}$, it was necessary to take the derivative of the output of the pitch rate gyro, which was accomplished by means of a washout circuit. It was necessary that the washout circuit contain a relatively large time lag and it was also necessary to provide a small amount of lead on the accelerometer signal. This was accomplished without increasing the order of the model transfer function by using a lead-lag network with the same lag as that of the washout circuit. The resulting configuration of the feedback instrumentation is shown in the following sketch:



This configuration resulted in a feedback transfer function x/A_N which was equivalent to a model transfer function of the form:

$$\frac{A_N}{A_{N,c}} = \frac{\tau_0 s + 1}{\left(\tau_2 s + 1 \right) \left(\frac{s^2}{\omega_0^2} + \frac{2\zeta_0}{\omega_0} s + 1 \right)}$$

If it were possible to filter the noise from the output of the normal accelerometer in the correct manner, it would be possible to design a system which would operate satisfactorily with two differentiations of the accelerometer output, and the invariant model could be realized. Since no data were available on the noise spectrum of the accelerometer in flight, it was necessary to leave this consideration to further study. The system with the variable model involved most of the problems (other than this particular problem of filtering) which would be encountered in the design of a system with an invariant model.

Dynamic Characteristics of System Components

The frequency response of elevon deflection to a sinusoidal voltage input to the airplane pitch-damper system was measured. It was found that the damper system could be adequately represented by a single quadratic transfer function of 4.8 cycles per second natural frequency and critical damping. Nominal values of the dynamic characteristics of the measuring instruments given by the manufacturers were a natural frequency of 20 cycles per second and a damping ratio of 0.5 for the normal accelerometer and a natural frequency of 35 cycles per second and a damping ratio of 0.7 for the pitch rate gyro. Because of the similarity of their dynamics, the feedback instruments were approximated in the preliminary design by the transfer function of the normal accelerometer.

Prediction of Chatter Frequency

The assumed instrument transfer function was used in combination with that of the airplane pitch damper and the approximate method of reference 2 to predict a chatter frequency of 3.9 cycles per second. This frequency was felt to be too low for a piloted airplane. Minor modifications to the pitch damper circuit resulted in sufficient improvement in the damper response to raise the predicted chatter frequency to 6.8 cycles per second, which was considered to be satisfactory.

Design of the Model

The design of the model for the present system was considerably more complicated than for the missile of reference 2 for the following reasons:

1. The necessity for using a washout circuit on the output of the rate gyro meant that the model transfer function had to be third order.

2. The variations in model characteristics with flight condition were such that a model which had a good response at the lowest dynamic pressure was considered unacceptably abrupt for a piloted aircraft at the highest dynamic pressure, whereas one need consider only structural limitations in the case of the missile application (ref. 2).

3. Inclusion of the transonic region resulted in the airplane's having minimum damping at the highest natural frequency. The complications resulting from this fact will be discussed later.

4. The numerator terms in the airplane A_N/s transfer function were not negligible in the flight regime being considered and had to be taken into account when the feedback quantities were computed.

Since the aircraft was to be piloted, the ideal model was chosen as one whose response was considered desirable by pilots - namely, a second-order transfer function with a natural frequency of 3.5 radians per second and a damping ratio of 0.6. These values were selected from the high-frequency region of "good" characteristics of reference 4 and fell within the low-frequency portion of the "best tested" area of reference 5. The feedback quantities were then chosen so as to produce this model at an intermediate value of dynamic pressure ($M = 1.15$ at $h = 30,000$ ft). Examination of the system for the highest dynamic pressure, however, showed that the very low aircraft damping coupled with the high natural frequency produced a condition wherein the combined phase shift of the airplane, pitch damper, sensing instruments, and integrator was about 172° . It was found that by designing for a higher natural frequency and lower damping for the second-order factor and adjusting the value of τ_0 properly a better phase margin could be provided and the response at low dynamic pressures could be made faster.

Control of Chatter Amplitude

During the frequency response tests of the pitch damper system, it became evident that fatigue due to structural resonance, and not the response of the airplane to the chatter, might be the limiting factor on the chatter amplitude. Opinions of the airframe manufacturer, based on flutter and vibration data, indicated that the following restrictions on elevon chatter should be observed:

Frequency < 10 cycles per second

Amplitude $< \pm 0.1^\circ$

It was decided to design for a chatter frequency of 40 radians per second (6.4 cps) to avoid the elevon resonance and to restrict the amplitude to

$\pm 0.05^\circ$ to provide an added safety factor. To accomplish this end, filters were placed before and after the limiter in the forward loop. The general requirements for design of these filters were as follows:

1. The total open-loop phase lag of the system, without the integrator, should be considerably less than 90° up to a frequency of 40 radians per second and should then change abruptly to more than 90° .
2. The gain of the filter preceding the limiter should be kept as low as possible at the higher frequencies in order to minimize reduction of limiter gain due to noise (including the chatter).
3. The filter following the limiter must provide sufficient attenuation to reduce the elevon motion at the chatter frequency to the desired level.

The design was accomplished by successive approximation using Bode plots. The choice of constants was influenced by the fact that the filters were to be constructed using operational amplifiers and R-C networks, and it was desired to use only components which were readily available. Figure 3 shows the phase curves of the added filters and of the filters combined with the damper and sensing instruments. Figure 4 shows the amplitude ratio curves of the filters preceding and following the limiter and the combined attenuation of the integrator-filter and damper following the limiter.

Ground Simulation

Before flight test, the system was simulated on an analog computer with the actual pitch damper system (including the hydraulics and elevon surfaces), amplifiers, and networks to be used in flight included in the loop.

It was found that the transient portions of responses to step commands were unsymmetrical at the high dynamic pressures; that is, much too abrupt in one direction. It turned out that this could be virtually eliminated by moving a portion of the filter preceding the limiter into the feedback path. This had about the same effect as passing the command signal through a small time lag and caused a small delay in the response to input commands.

The final configuration of the system and the resulting model are shown in figure 5. The small first-order lag factors $0.025 s + 1$ were included in the feedback networks for added stability. Time histories obtained on the ground of the output of the model in response to 2 g step commands at the five design flight conditions are presented in figure 6. The model in this case consisted of the following portion of the model transfer function given in figure 5:

$$\frac{A_N}{A_{N,c}} \approx \frac{(0.865 s + 1)}{\left(\tau_2 s + 1\right) \left(\frac{s^2}{\omega_0^2} + 2 \frac{\zeta_0}{\omega_0} s + 1\right)}$$

Similar responses of the over-all system (including the complete model) are shown in figure 7.

FLIGHT TESTS

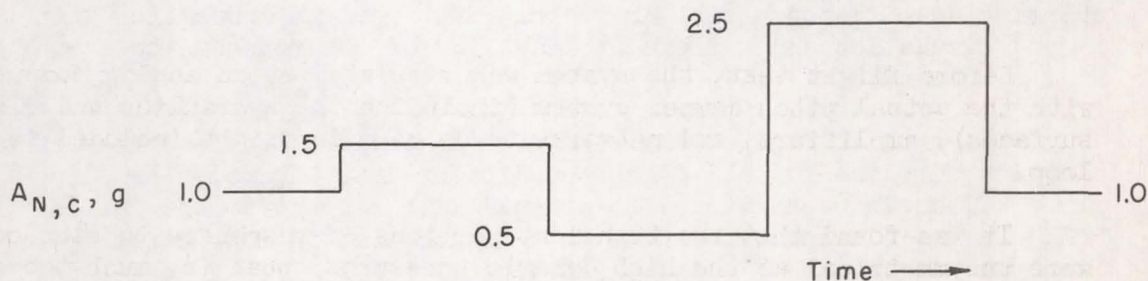
The initial part of the flight tests consisted of a documentation of the longitudinal response of the adaptive airplane to step ΔA_N commands, and of the basic (unaugmented) airplane to step elevator inputs, at the five flight conditions selected for the preliminary study (table I) and at three additional "off-design" conditions as a further check on the effectiveness of the system. These additional conditions were:

$M = 0.65$ at 10,000 ft

$M = 0.60$ at 25,000 ft

$M = 0.85$ at 25,000 ft

The step commands were obtained by means of a specially designed function generator whose output varied in the following manner:



The steps either could be set to occur automatically at time intervals of 5 seconds or could be introduced manually by the pilot. The purpose of the alternating positive and negative feature mainly was to obtain steps of various amplitudes without developing excessive pitch angles during the runs.

The second phase of the flight tests was devoted to a pilot evaluation of the system from the standpoint of the adaptive airplane response to pilot-introduced normal-acceleration commands through the side controller, both in pull-up and push-down maneuvers and in abrupt turn entries. At

the same time, characteristics such as chatter frequency and amplitude and adaptive airplane response to turbulent air were monitored and evaluated by the pilot.

RESULTS AND DISCUSSION

Adaptive Response

Time histories of typical flight runs made at $M = 1.15$ and $h = 30,000$ feet and at $M = 0.80$ and $h = 40,000$ feet are presented in figure 8. Shown are pitching velocity, normal acceleration, and elevator responses of the adaptive airplane to the normal-acceleration commands applied by the step-function generator. Included on the same figure for comparison are responses of the basic airplane to step elevator commands for corresponding flight conditions. It should be noted that for the adaptive airplane the $\Delta A_N, c$ inputs were the same magnitude at all flight conditions, while for the basic airplane the elevator angle commands were standardized. Consequently, the basic-airplane responses of figure 8 were scaled in amplitude to correspond with the $\Delta A_N, c$ step commands to the adaptive airplane.

It is apparent from figure 8 that the adaptive system was successful in eliminating the large overshoots resulting from the relatively low damping of the basic airplane; in fact, at $M = 1.15$ and $h = 30,000$ feet, the adaptive airplane response appears to be more heavily damped and sluggish than would be desired. Also apparent in figure 8 is a delay in adaptive A_N response similar to that mentioned in connection with the ground simulation and a frequent inability of the response to steady out at the commanded value.

Presented in figure 9 are the normal-acceleration responses of the adaptive airplane for all flight conditions investigated, including the three off-design conditions, normalized with respect to the initial peak $|\Delta A_N|$ value. It can be seen in this figure that in most cases no reliable steady-state values of ΔA_N existed until fairly late in the runs, if at all. Therefore, during the analysis, attention was given mainly to the first part of the response.

The undamped natural frequency and damping ratio in pitch of the basic airplane and of the adaptive airplane are compared in figure 10 as functions of Mach number and altitude. The values given in this figure for the adaptive airplane are those of second-order responses which appear best to approximate the first 2 to 4 seconds of the responses measured in flight. The method used to fit the second-order responses to the adaptive responses is outlined in appendix B. The actual response of the adaptive airplane was similar to the third-order output of the model.

As in the case of the basic airplane, the natural frequency of the adaptive airplane increased with speed at a given altitude; however, the degree of increase was considerably smaller. In comparison with the rapid decrease of the damping of the basic airplane with increasing speed, the damping ratio of the adaptive airplane (except at $M_1 = 1.15$ and $h_p = 30,000$ feet) is seen to remain between 0.56 and 0.69. These values are within the region considered good by pilots (refs. 4 and 5) and, especially at the higher speeds, represent a substantial improvement over the damping of the basic airplane. The response at condition 3 was best fitted by an overdamped second-order response with a damping ratio of 1.7 and a natural frequency of 4.00 radians per second.

A The natural frequency of the adaptive airplane is compared in figure
5 11 with the natural frequency of the basic airplane at all flight condi-
1 tions investigated. This figure indicates, in a different way than does
0 figure 10, the reduction in spread of the longitudinal frequency over
the flight envelope. Apparent also from figure 11 is a fairly smooth
and progressive variation of adaptive airplane frequency with basic
airplane frequency.

In figure 12, the natural frequency and damping ratio of the adaptive airplane in flight are compared with the characteristics of the adaptive airplane as simulated on the ground (from fig. 7). The ω_n and ζ of the analog curves were estimated by the same method applied to the flight data (appendix B). Figure 12 shows, except for $M = 1.15$ and $h = 30,000$ feet, reasonable agreement between the ground simulation results and the flight results.

System Chatter

Measurements of the chatter frequency showed a variation in flight from 8.9 to 9.5 cycles per second, somewhat higher than the design value of 6.4 cycles per second (40 radians per second). This increase in chatter frequency was probably due to the sensing instruments' having a faster dynamic response than assumed in the calculations. The percentage variation of chatter frequency was not large and did not appear to be a function of any flight variable. Although the chatter frequency in flight was in the region of the elevon and wing resonant frequencies (about 10 cycles per second), the amplitude was not considered excessive by the pilots who performed the flight tests. The elevon chatter amplitude was estimated from the flight records to be $\pm 0.07^\circ$; the maximum resulting airplane normal-acceleration amplitude was in the region from ± 0.025 g to ± 0.035 g at the highest q condition, $M = 0.94$ at $h = 12,000$ feet.

The chatter amplitude at the center stick was sufficiently large to be noticed by the pilots. The project pilot did not object to the chatter at all; it was not noticed by him in the airframe motion. One other pilot who participated considered the chatter to be slightly objectionable.

Behavior in Rough Air

The flight records and pilot comments indicated that the system was very sensitive to disturbances encountered while flying in rough air. While the system was quick to return the airplane to the commanded A_N level following a gust disturbance, the elevon deflections resulting therefrom caused exaggerated changes in pitch angle which were extremely disconcerting to the pilots and made any kind of tracking task nearly impossible.

The sensitivity to rough air may have been aggravated by the high open-loop gain and the low damping of the quadratic mode of the model.

Behavior Under Pilot Control

A time history of airplane motions in response to pilot control through the side stick is presented in figure 13. The maneuver performed was a forced longitudinal oscillation followed by a turn to the left and recovery to level flight. The delay in the A_N response which was mentioned earlier, and to which the pilots objected, is apparent in the figure.

It was noted that at the lower speeds the pitch attitude response to a given motion of the side stick was unusually large and annoying. This was true especially at the slowest flight condition ($V_1 = 200$ knots; $h = 10,000$ feet). This behavior was due to the fact that the pilot was controlling in pitch through a "g-command" system rather than the usual attitude-command system.

General Observations

The relatively constant output feature of the adaptive control system was considered by the pilots to be desirable; however, the benefits of the system tended to be minimized by the fact that over the flight envelope of the tests, the variation in response of the basic airplane with constant-gain dampers was not very noticeable to the pilots.

During the design of the system, it became apparent that by using high-performance components it would be possible to build a linear system with constant gains which would operate satisfactorily at all flight conditions investigated, except at the highest dynamic pressure. This suggests that the designer might seriously consider a system with a single step gain change at a predetermined value of dynamic pressure or, in the case of an airplane of a lower speed range, a linear system with constant coefficients.

According to pilot comments, the A_N command feature of the system made it possible for the airplane to maintain nearly constant altitude while decelerating through the transonic speed range; that is, the transonic trim change was automatically corrected.

CONCLUDING REMARKS

A normal-acceleration-command adaptive control system has been designed, ground tested by means of an analog computer, and flight tested in a piloted airplane; the analog and flight test results were in agreement.

The system maintained nearly constant damping of the airplane response to step normal-acceleration commands. Except for the supersonic case, which was overdamped with the system operating, the damping ratio was maintained between 0.56 and 0.69. These values represent a considerable improvement over the basic damping of the airplane and are in the region considered good by pilots.

The natural frequency of the adaptive airplane varied from 1.70 to 4.23 radians per second. This variation was about half that of the basic airplane.

The normal-acceleration command feature of the system automatically compensated for the transonic trim change.

The system chatter, having a frequency of about 9 cycles per second and an amplitude between $\pm 0.05^\circ$ and 0.10° of elevon (resulting in a maximum airplane normal-acceleration response of from ± 0.025 g to ± 0.035 g), was marginally detectable and barely objectionable to the pilots. Such chatter was not always present.

Accentuated pitch attitude responses to gust disturbances and to pilot inputs, together with a delay in normal-acceleration response to pilot control, would tend to limit the use of the present system in routine piloted flight; however, use of such a system for certain specialized tasks, such as during the attack mode of an automatically controlled interceptor, seems feasible.

Ames Research Center

National Aeronautics and Space Administration

Moffett Field, Calif., Feb. 21, 1961

APPENDIX B

METHOD USED TO DETERMINE NATURAL FREQUENCY AND DAMPING
OF THE ADAPTIVE AIRPLANE

The equivalent undamped natural frequency and damping ratio at each flight condition of the test airplane with adaptive system operating were taken as those characteristics of second-order step responses which best approximated the third-order responses measured in flight. The normalized $|\Delta A_N|$ step responses chosen for analysis are presented in figure 14. Each solid curve shown represents the faired average of the best-defined responses at each flight condition.

While the flight data at some conditions (figs. 14(a), (c), and (f)) were readily fitted by second-order step-response curves, the data at the remaining conditions showed a marked decay of the response with time after a steady-state value should have been reached. Since no explanation for this behavior was apparent, some means of adjusting the data to a form which could be fitted was necessary.

The first step was to assume that each response originated at the minimum point of the small initial dip following the step command. The resulting curve was then multiplied by a function $(1 + Gt)$ where, in each case, G was chosen so that the product as a function of time would have a slope of zero at a time equal to one oscillation period after the initial minimum. The value of the product at this time was taken as the steady-state value of the adjusted response from which the damping ratio was determined by measurement of the initial overshoot (fig. 14).

Once the damping ratio was established in each case, the natural frequency was fixed by matching the value of the adjusted response when it first reached 80 percent of its maximum value (with the exception of $M = 1.15$ at $h = 30,000$ feet). The flight response at $M = 1.15$ and $h = 30,000$ feet appeared to be fitted best by an overdamped response curve of $\omega_n = 4.0$ radians per second and $\zeta = 1.7$.

The adjusted flight responses and fitted second-order curves are compared in figure 14.

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TABLE I.- AIRPLANE PARAMETERS USED IN DEVELOPMENT OF THE
ADAPTIVE CONTROL SYSTEM

Condition	1	2	3	4	5
V_i	200	---	---	---	---
M	0.36	0.94	1.15	0.80	0.96
h	10,000	12,000	30,000	40,000	40,000
$C_{L\alpha}$	2.63	3.20	2.97	2.86	3.20
$C_{L\delta}$	1.04	1.17	.39	1.12	1.01
$C_{m\alpha}$	-.20	-.38	-.52	-.22	-.42
$C_{m\delta}$	-.41	-.61	-.24	-.59	-.44
C_{mq}	-.78	-.35	-1.51	-1.23	-.21
$C_{m\dot{\alpha}}$	-.20	.05	.81	-.20	.34

$$S = 698$$

$$\bar{c} = 23.76$$

$$m = 746$$

$$I_Y = 126,000$$

TABLE II.- COEFFICIENTS OF AIRPLANE $\frac{A_N}{\delta}$ TRANSFER FUNCTION¹

$$\frac{A_N}{\delta} = -N \frac{1 + Ps + Qs^2}{s^2 + 2\zeta\omega_n s + \omega_n^2}, \text{ g/deg}$$

where

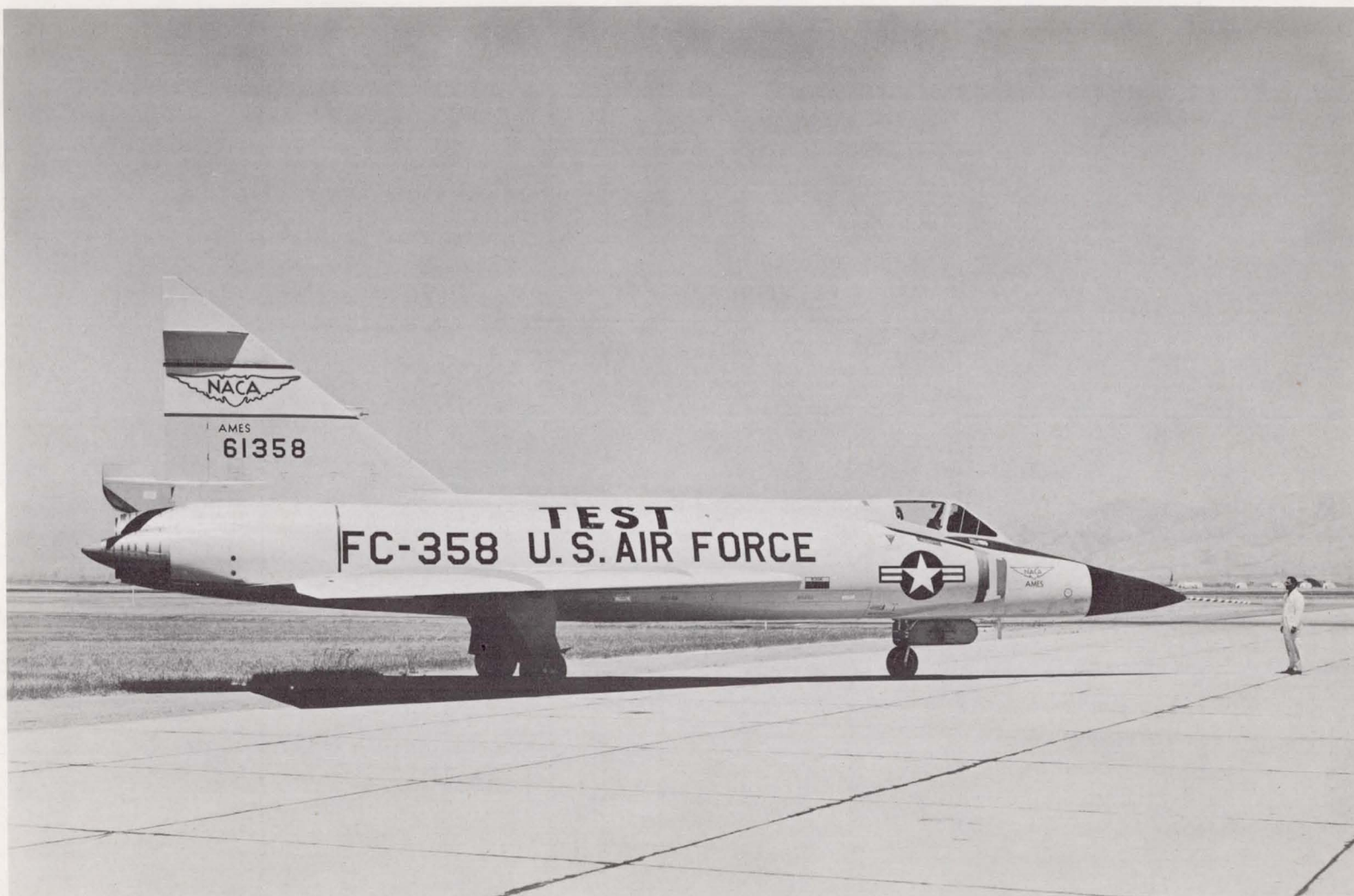
$$N = \frac{q^2 s^2 \bar{c} \left(C_{L_\delta} C_{m_\alpha} - C_{L_\alpha} C_{m_\delta} \right)}{m I_Y}$$

$$P = \frac{\bar{c}}{2V} \frac{C_{L_\delta} \left(C_{m_q} + C_{m_\alpha} \right)}{C_{L_\delta} C_{m_\alpha} - C_{L_\alpha} C_{m_\delta}}$$

$$Q = \frac{-I_Y C_{L_\delta}}{q s \bar{c} \left(C_{L_\delta} C_{m_\alpha} - C_{L_\alpha} C_{m_\delta} \right)}$$

Condition	N	P	Q	$2\zeta\omega_n$	ω_n^2
1	1.00	-0.0359	-0.0693	1.348	3.45
2	69.30	.0106	-.0071	.976	41.51
3	11.45	-.0056	-.0100	1.965	39.69
4	2.95	-.0170	-.0337	1.105	5.07
5	4.17	.0017	-.0310	.767	13.93

¹For derivation, see reference 3.



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Figure 1.- Airplane used in the present investigation.

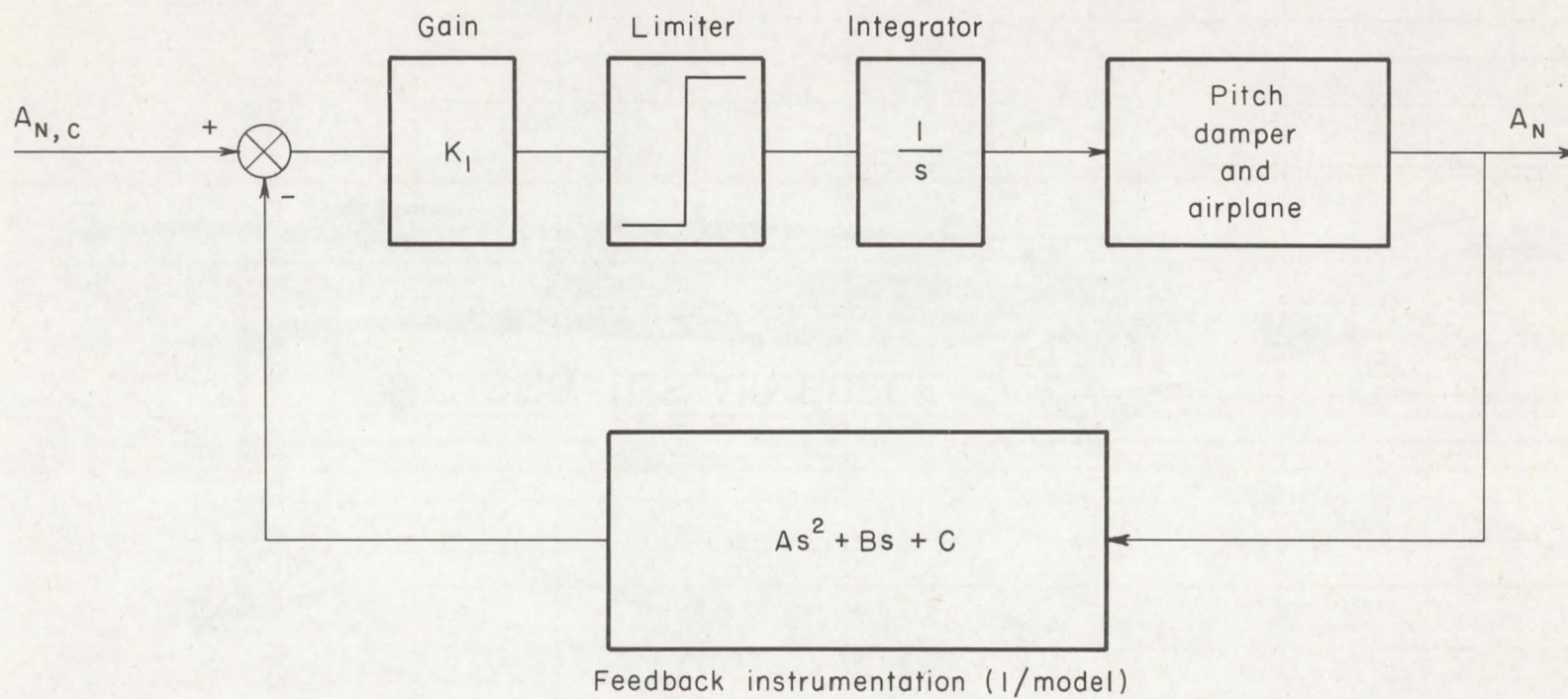


Figure 2.- Simplified block diagram of adaptive control system using normal-acceleration command.

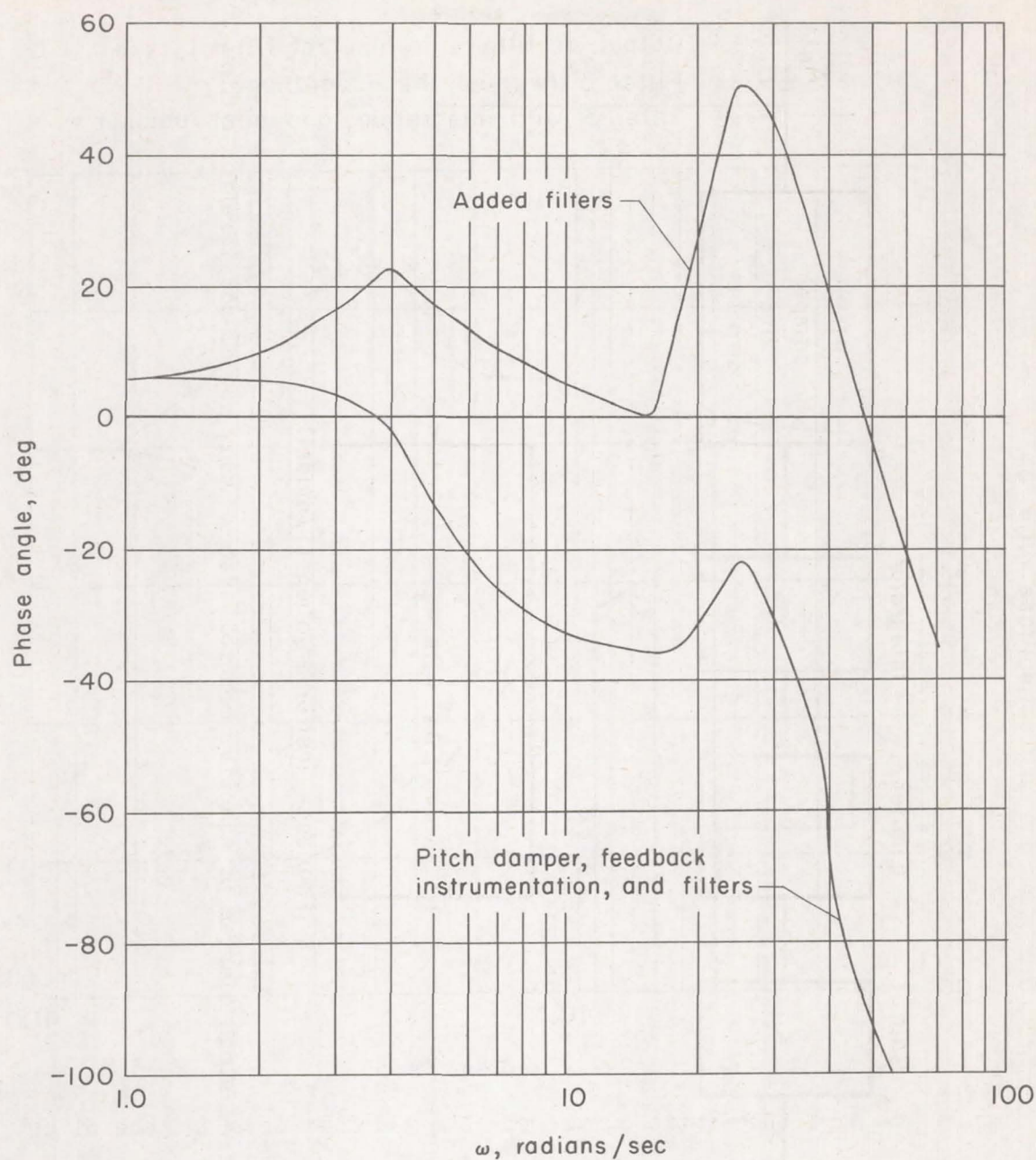


Figure 3.- Phase shift curves of the added filters, pitch damper, and feedback instrumentation.

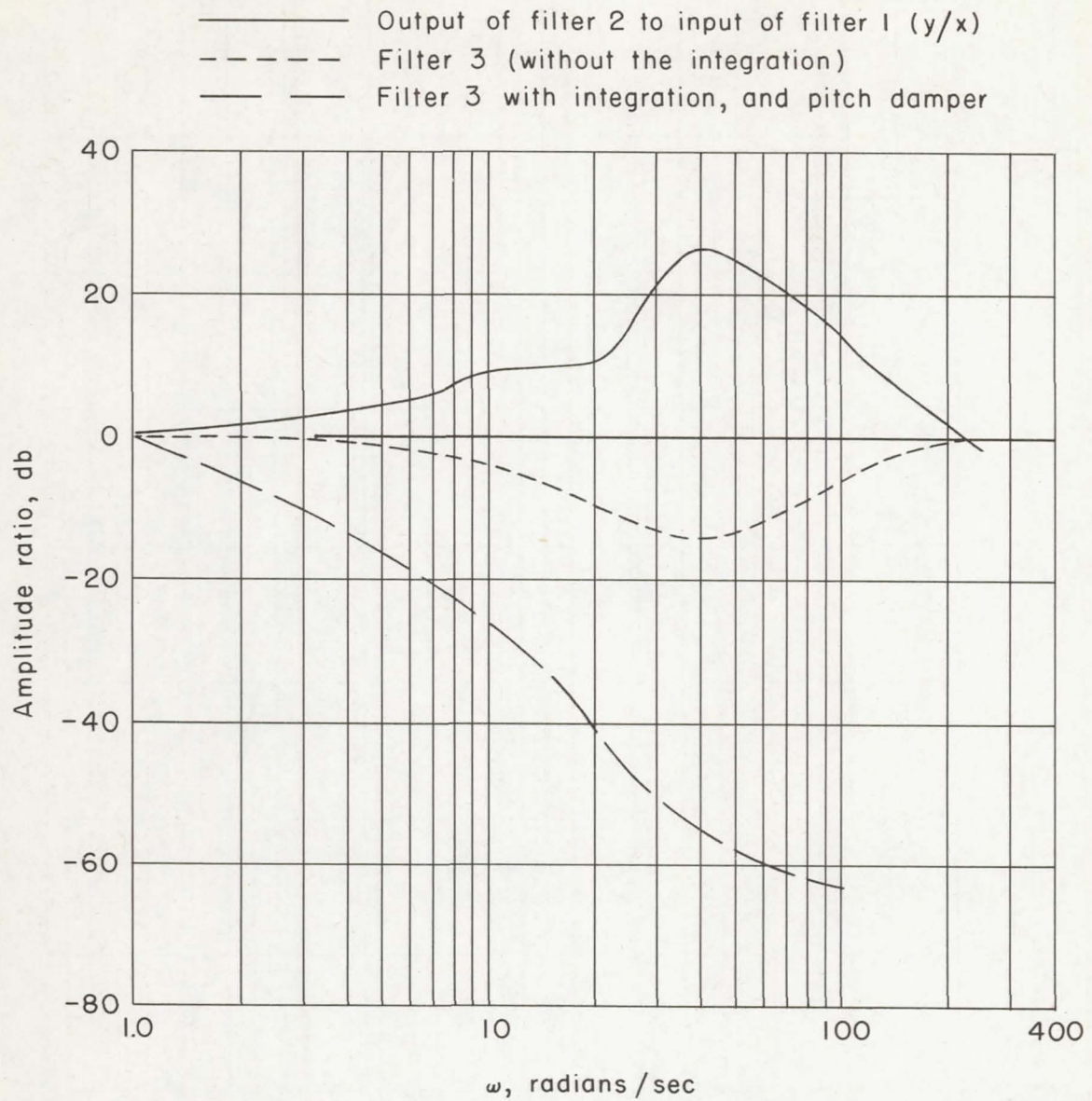


Figure 4.- Amplitude ratio curves for the added filters and the pitch damper.

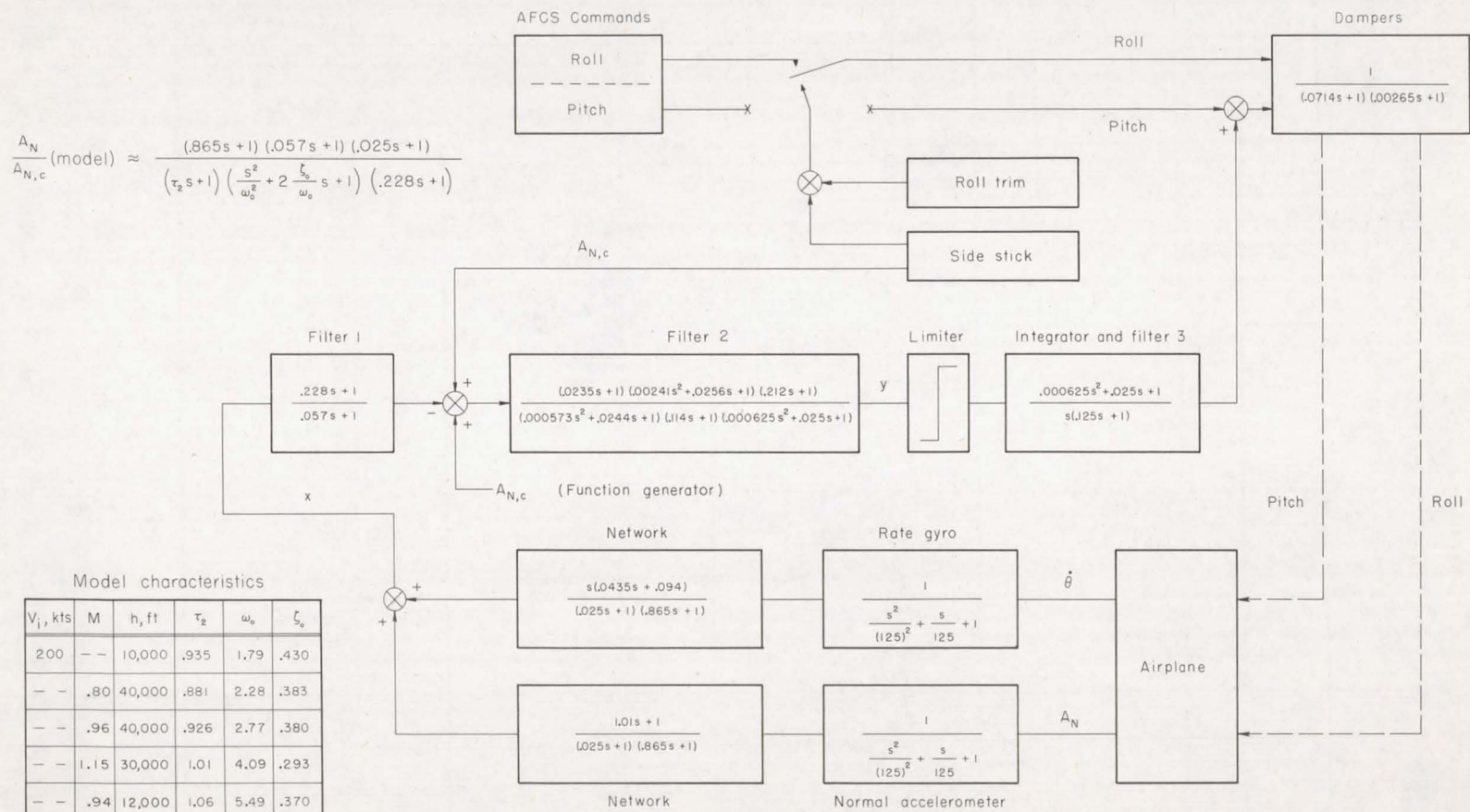


Figure 5.- Block diagram of final normal-acceleration adaptive system showing method of installation.

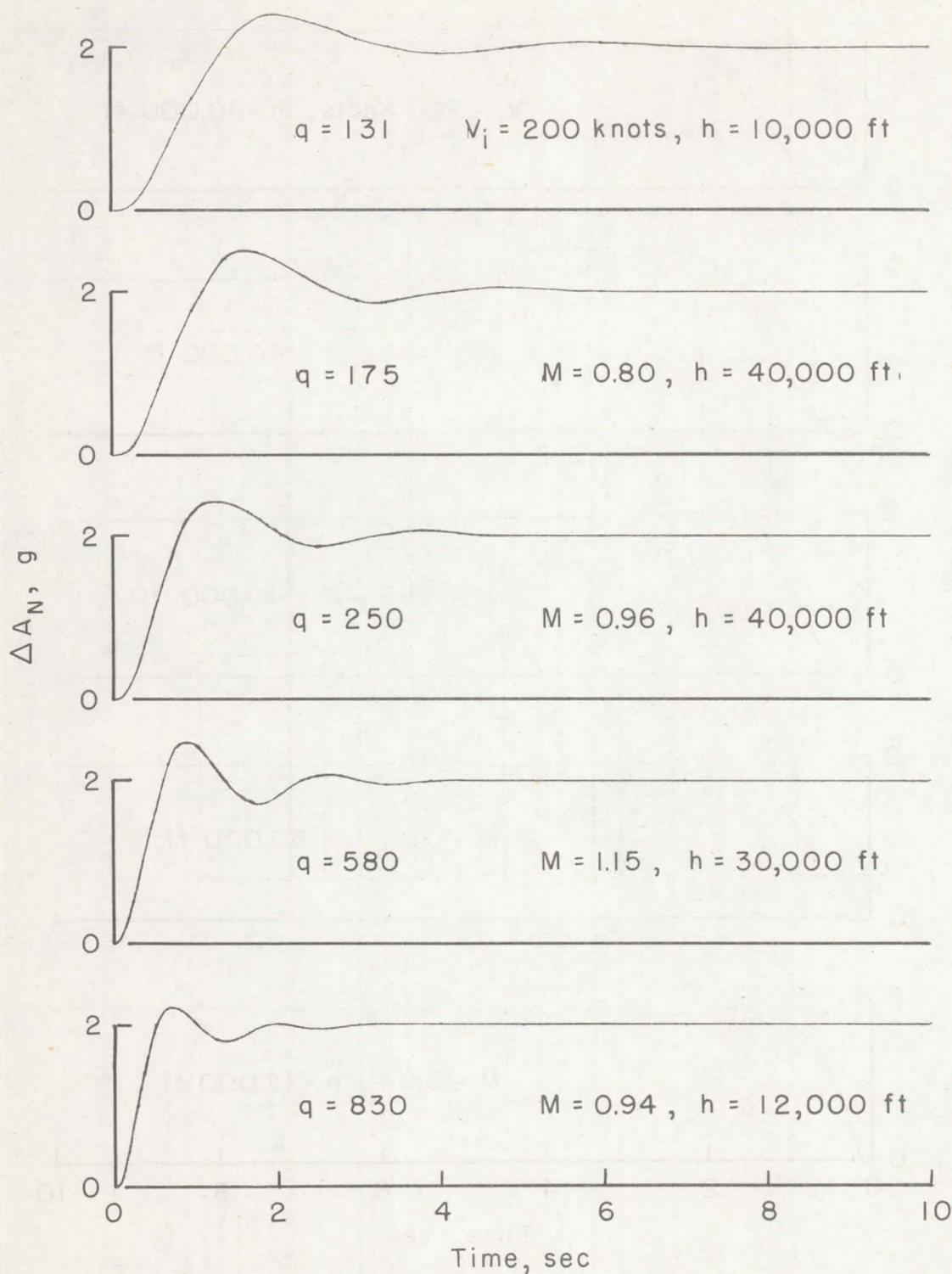


Figure 6.- Responses of final model to 2g step commands, arranged in order of increasing dynamic pressure.

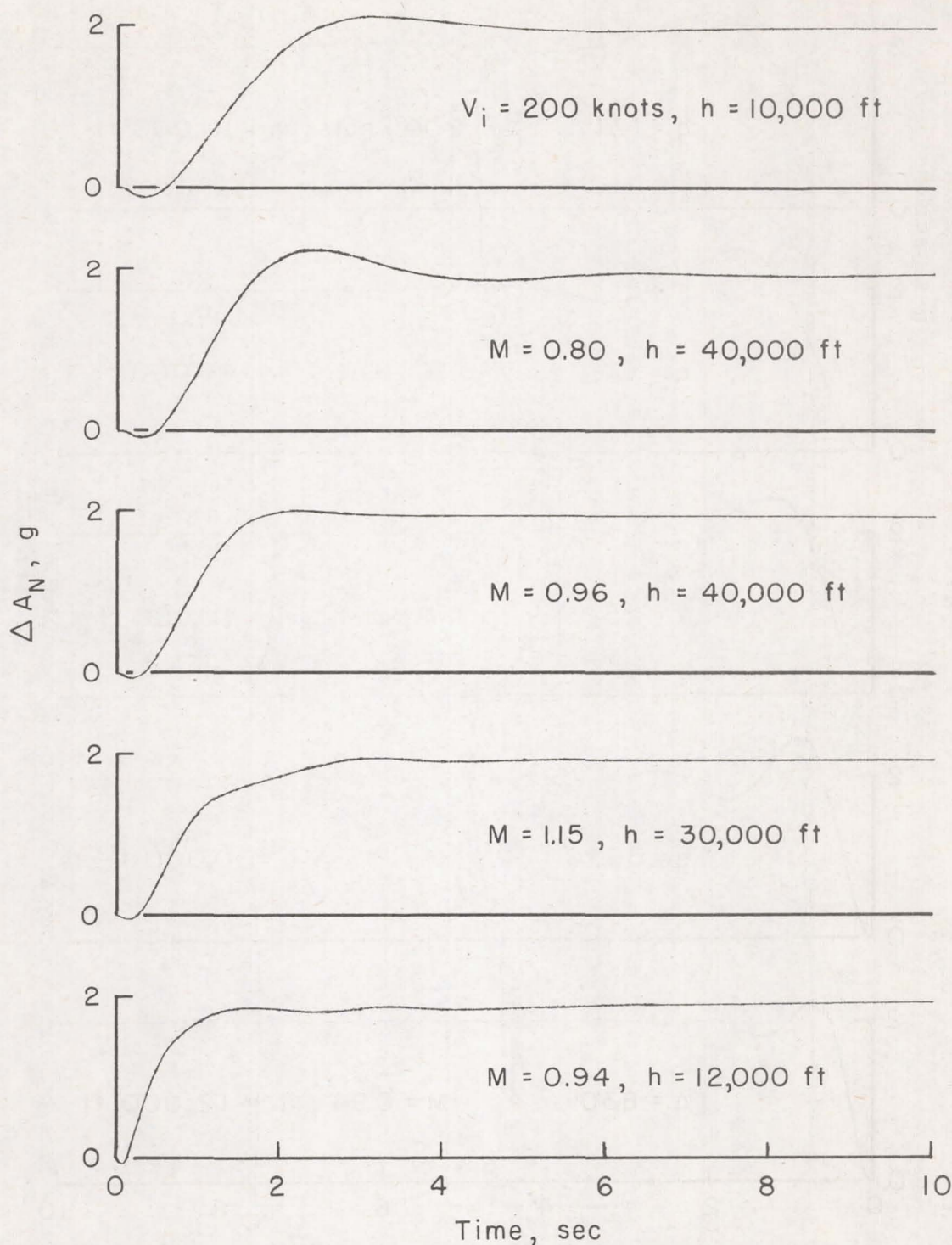
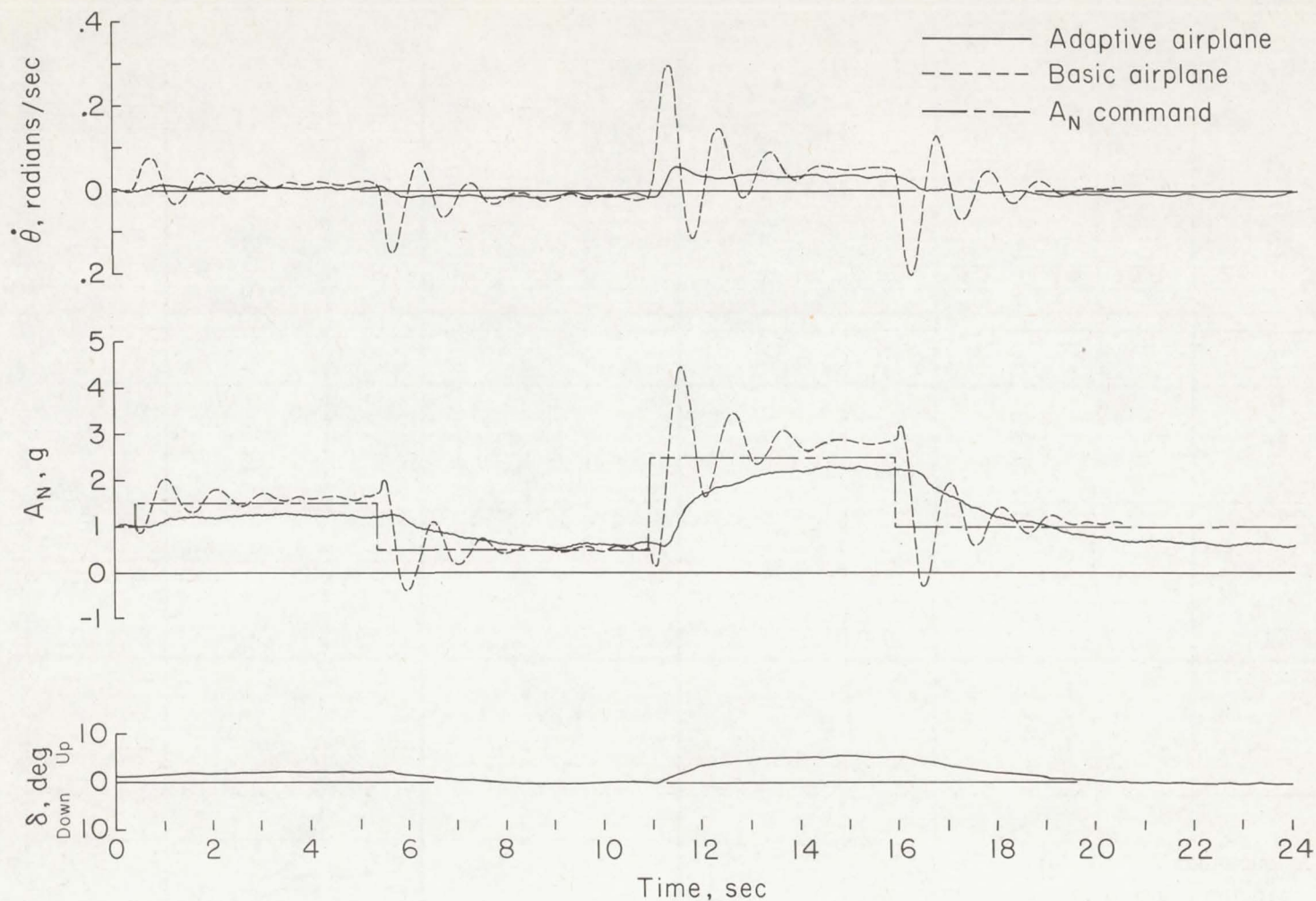
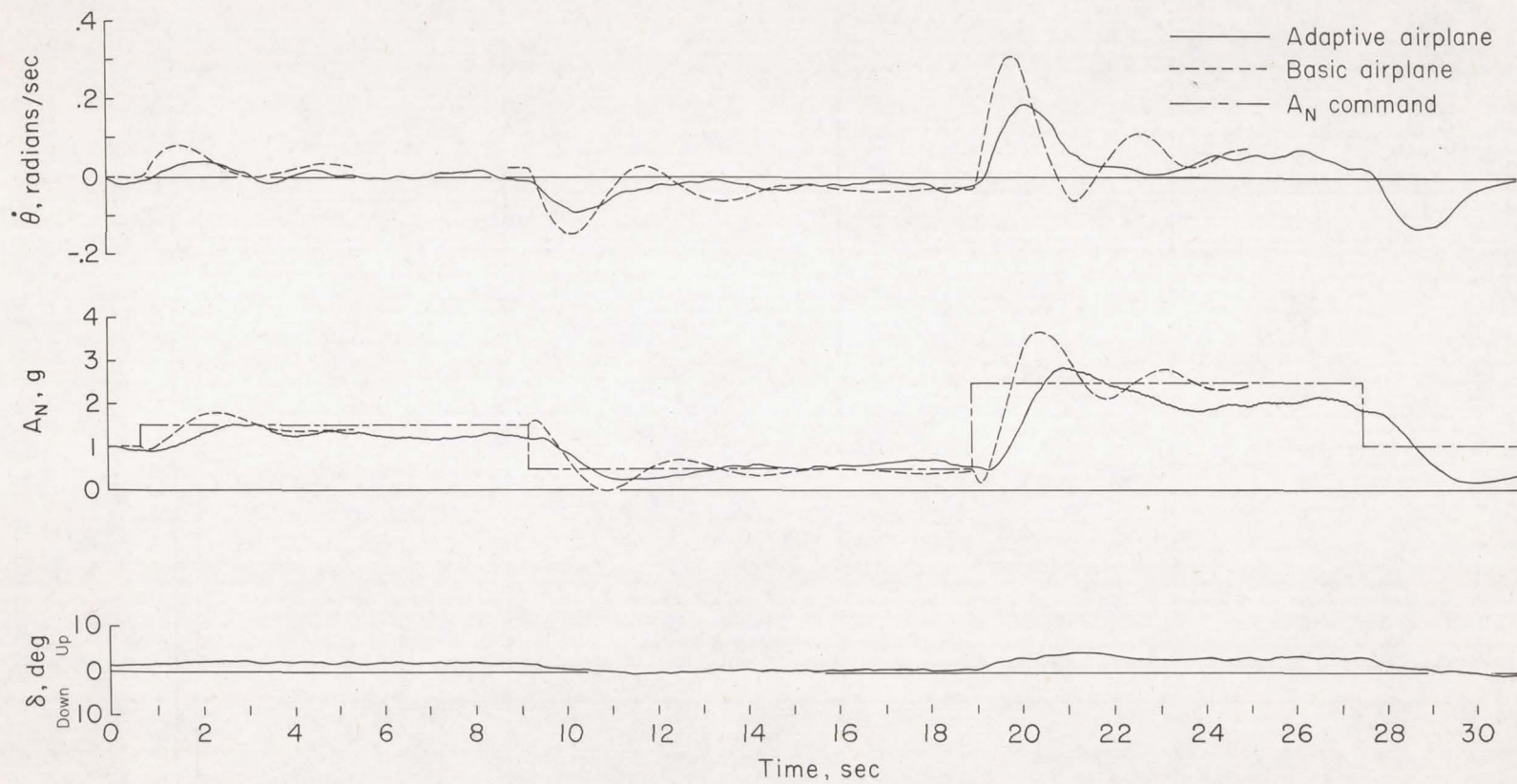


Figure 7.- Analog responses of adaptive airplane to 2g step commands, arranged in order of increasing dynamic pressure; final configuration.



(a) $M = 1.15$, $h = 30,000$ ft

Figure 8.- Time histories of typical basic and adaptive airplane $\dot{\theta}$ and A_N variations in response to step A_N commands.



(b) $M = 0.80$, $h = 40,000$ ft

Figure 8.- Concluded.

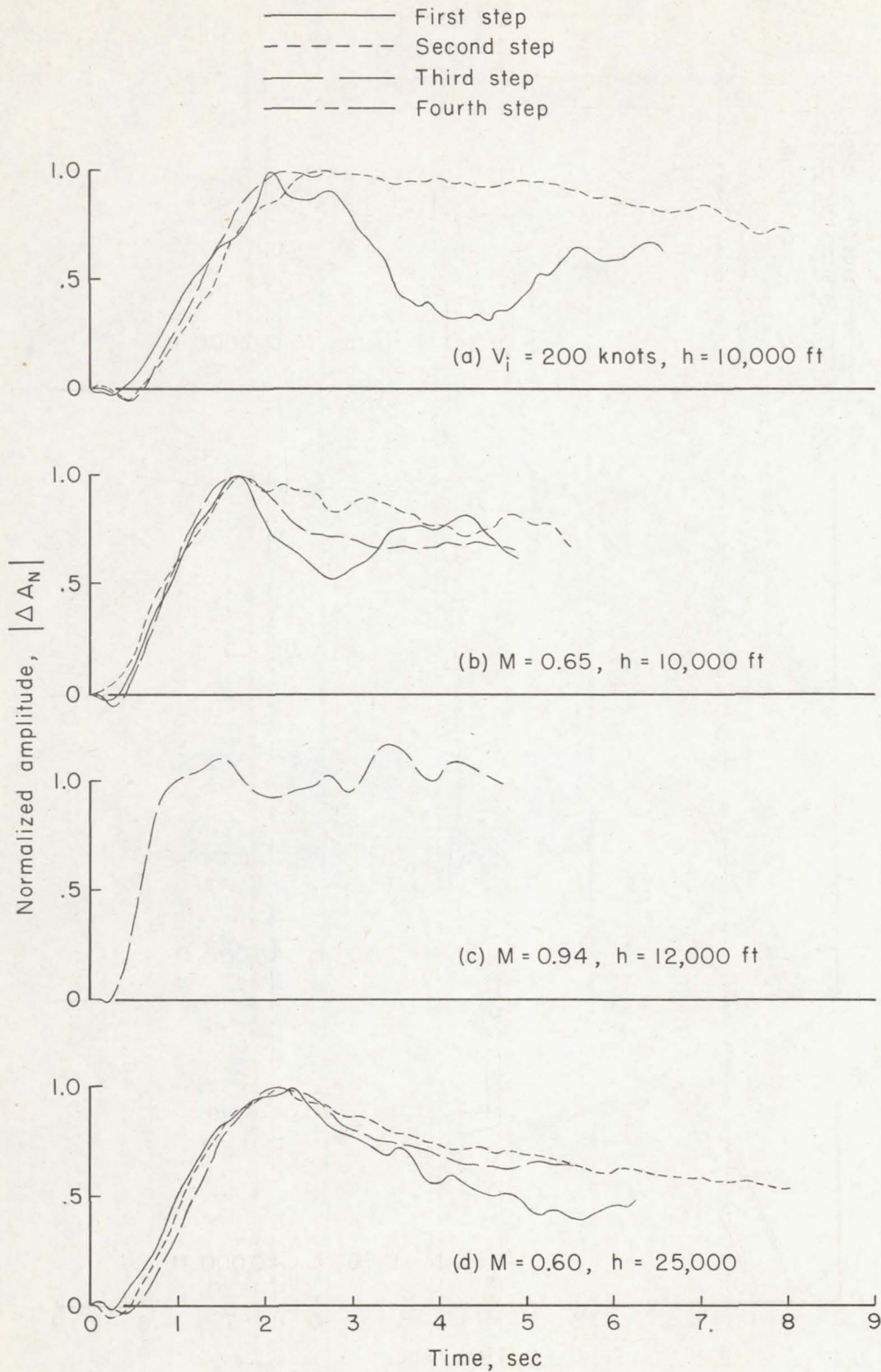


Figure 9.- Normalized $|\Delta A_N|$ responses of adaptive airplane to step commands from function generator.

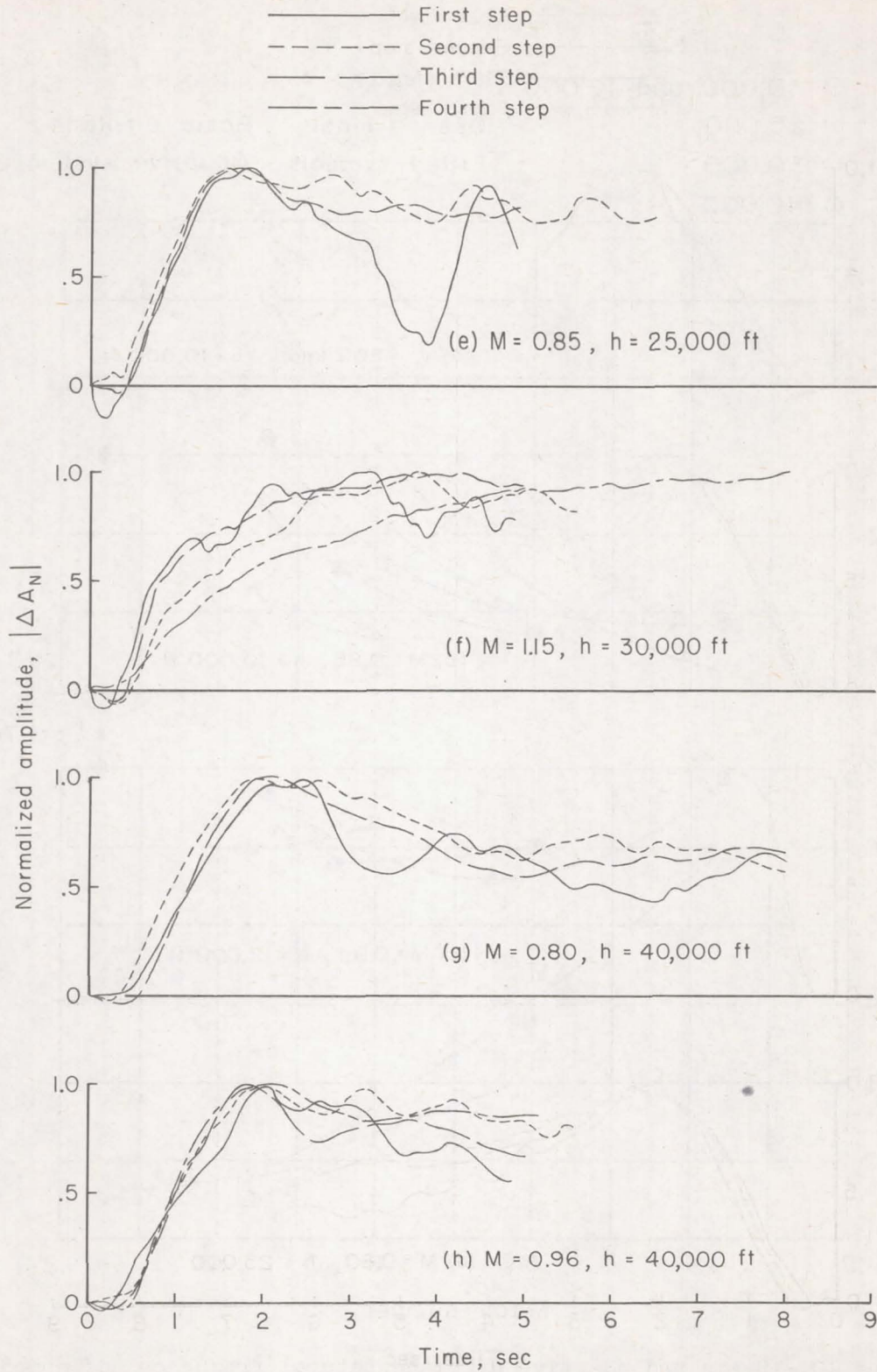


Figure 9.- Concluded.

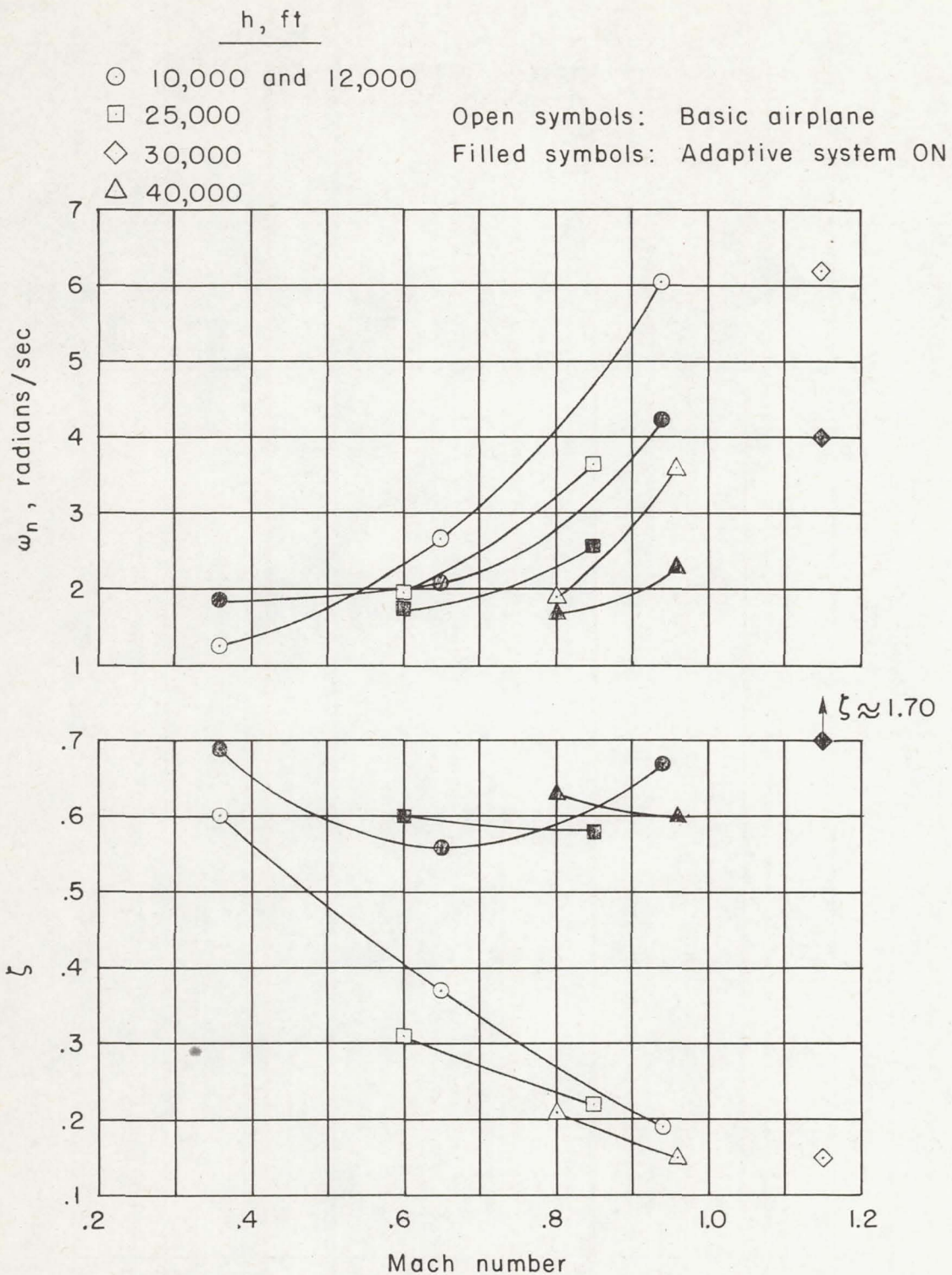


Figure 10.- Basic and adaptive airplane natural frequency and damping ratio as functions of Mach number for several altitudes.

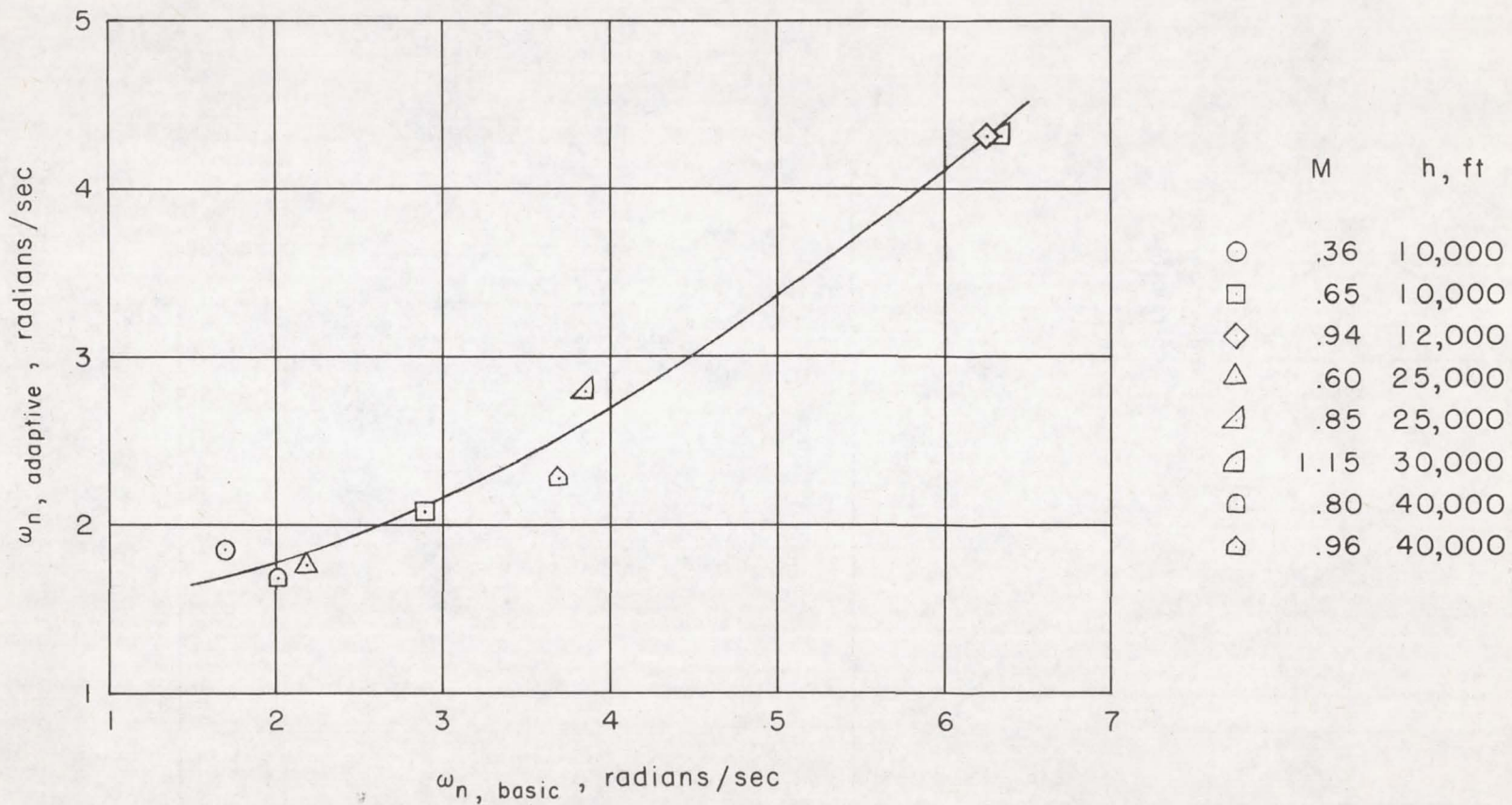


Figure 11.- Comparison between adaptive airplane natural frequency and basic airplane natural frequency for all flight conditions investigated.

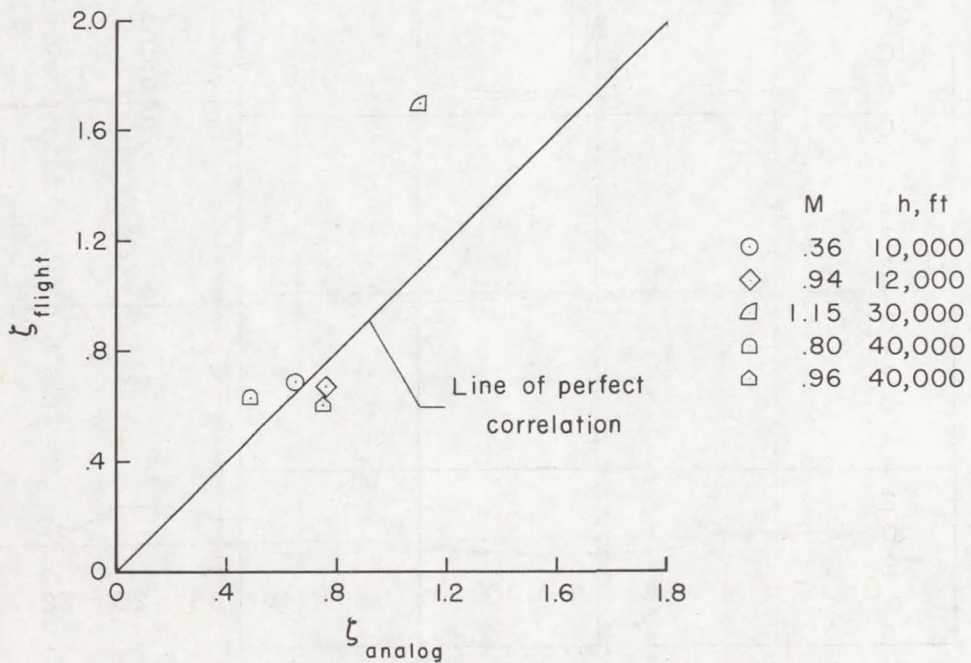
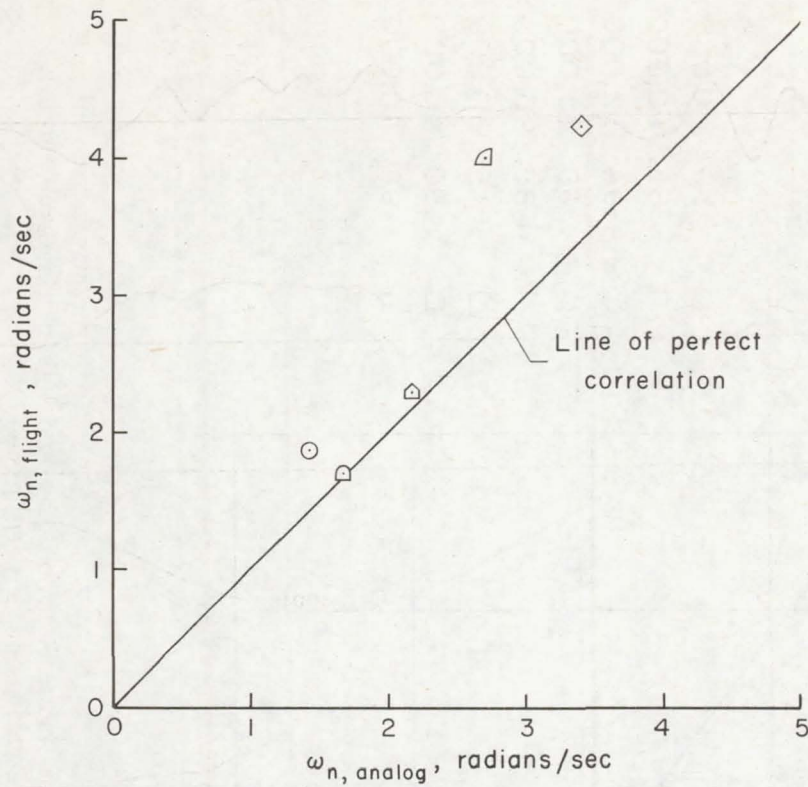


Figure 12.- Comparison between flight and analog values of natural frequency and damping of the adaptive airplane.

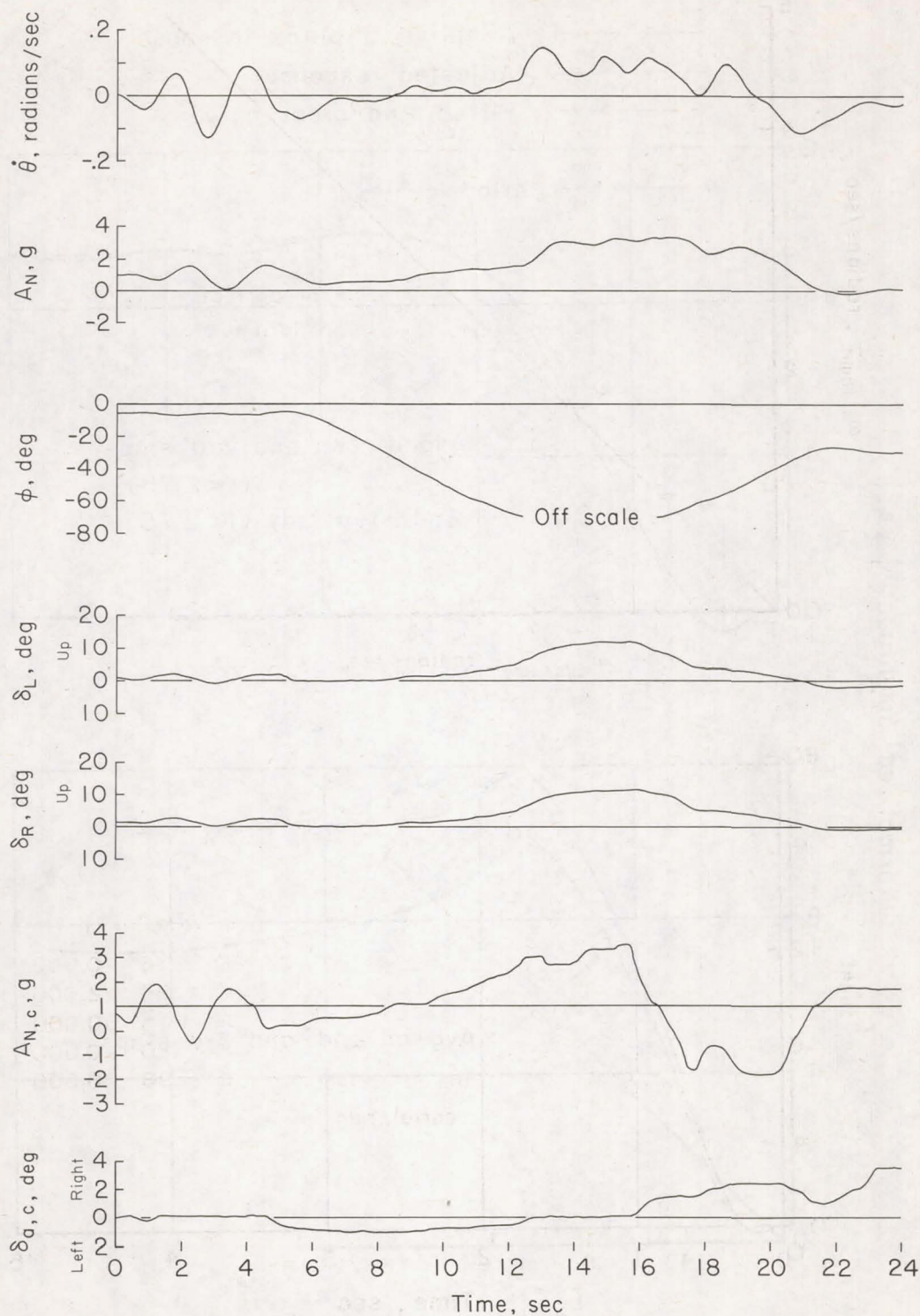


Figure 13.- Time histories of adaptive airplane motions in response to pilot control inputs through the side stick; $M = 0.96$, $h = 40,000$ ft.

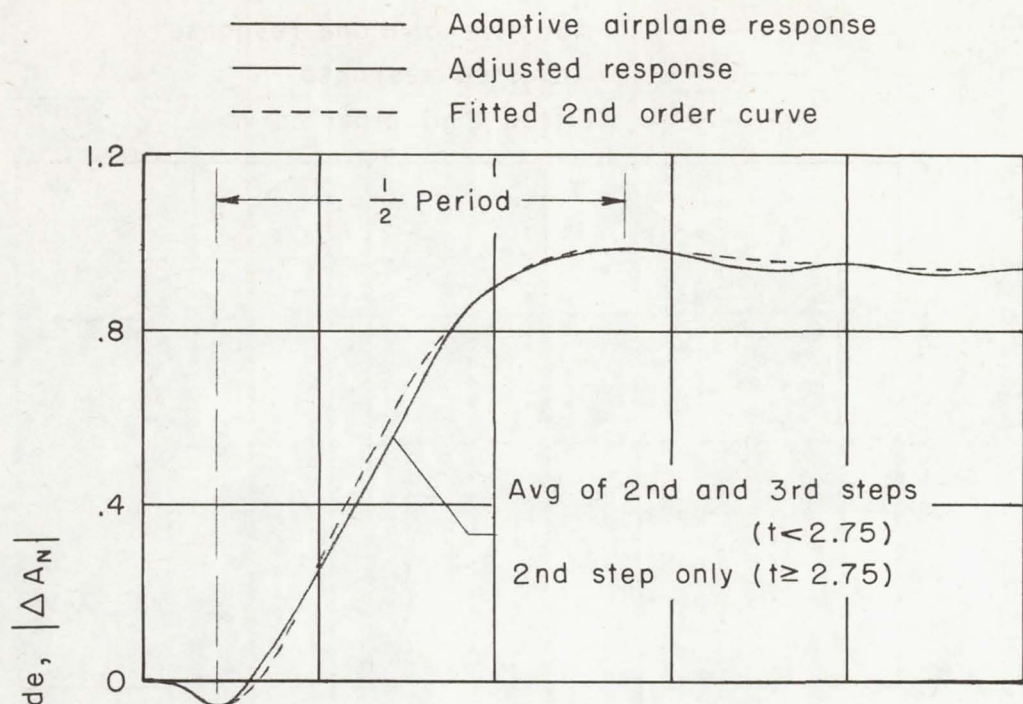
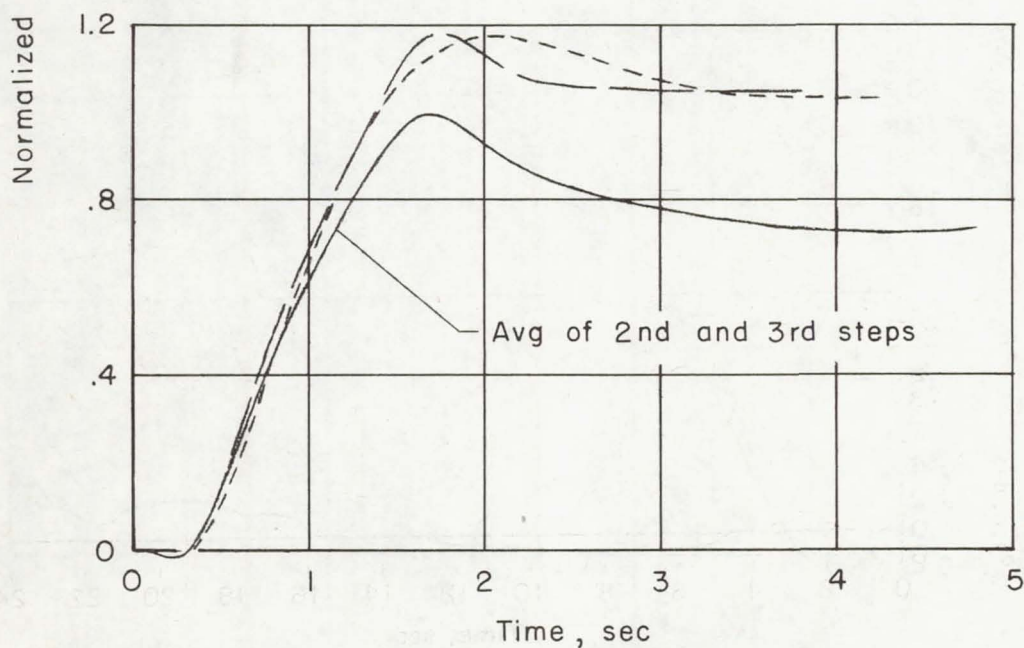
(a) $V_1 = 200$ knots, $h = 10,000$ ft(b) $M = 0.65$, $h = 10,000$ ft

Figure 14.— Average normalized $|\Delta A_N|$ responses of the adaptive airplane to step commands compared with fitted second-order curves.

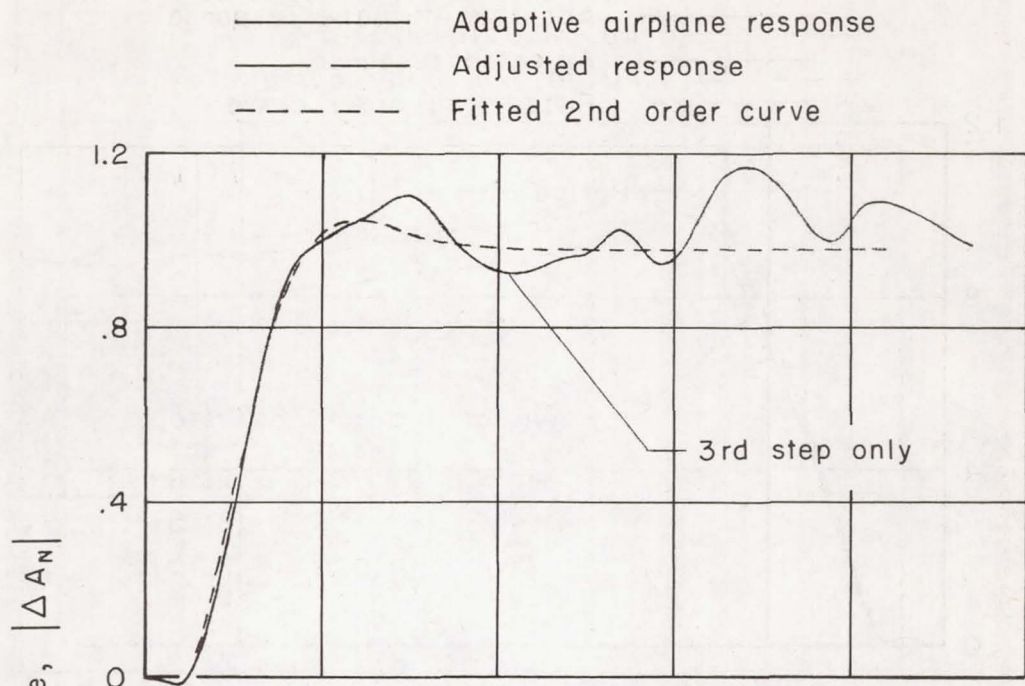
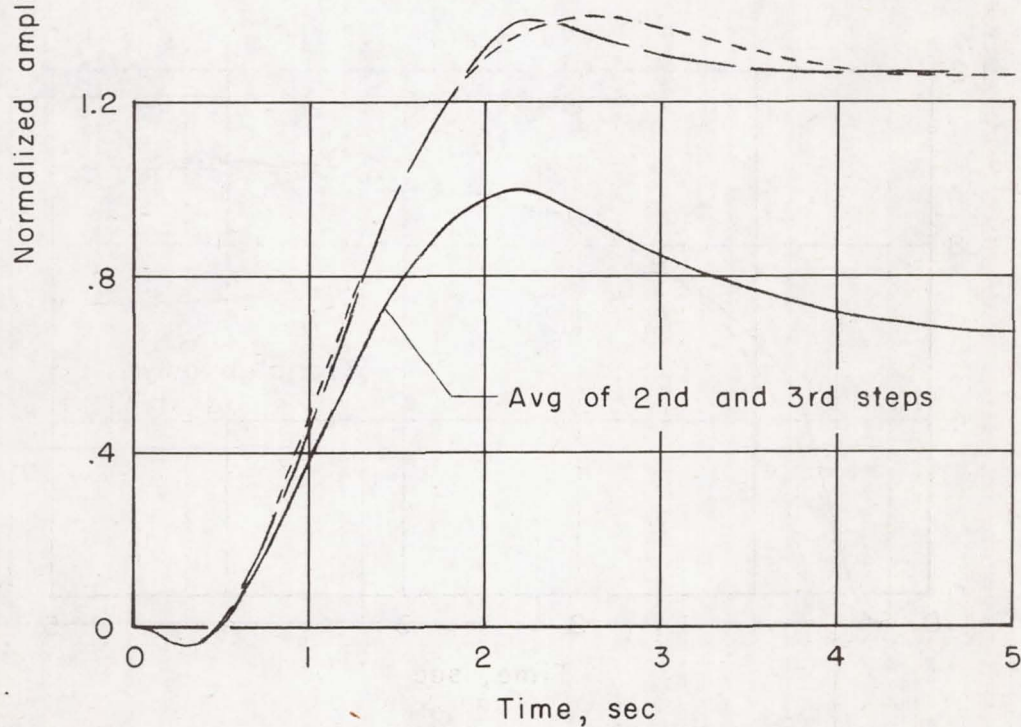
(c) $M = 0.94$, $h = 12,000$ ft(d) $M = 0.60$, $h = 25,000$ ft

Figure 14.- Continued.

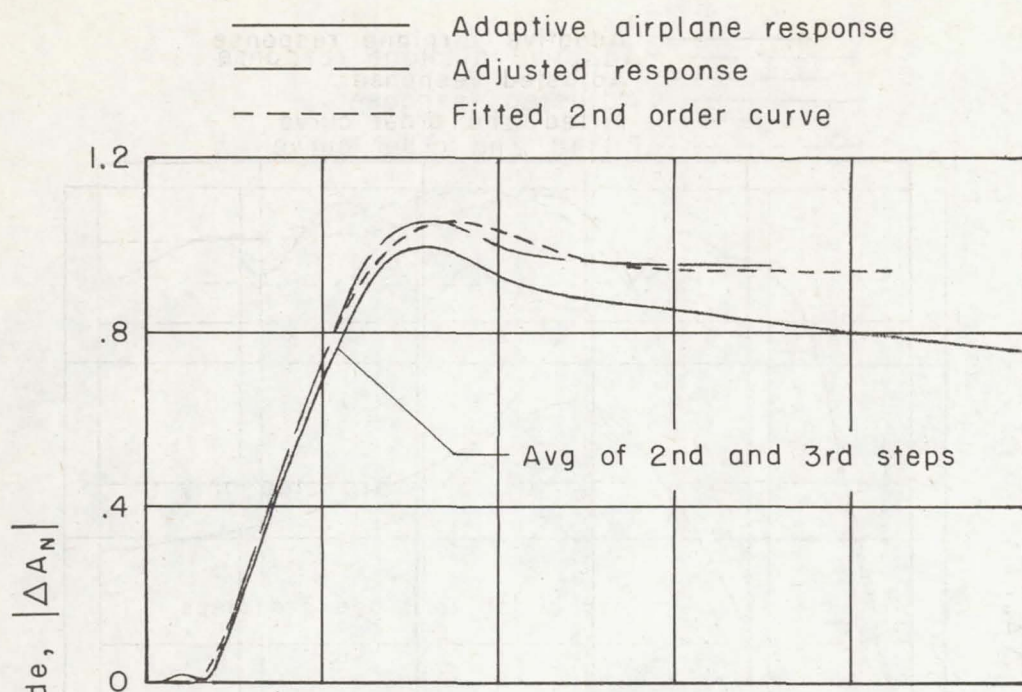
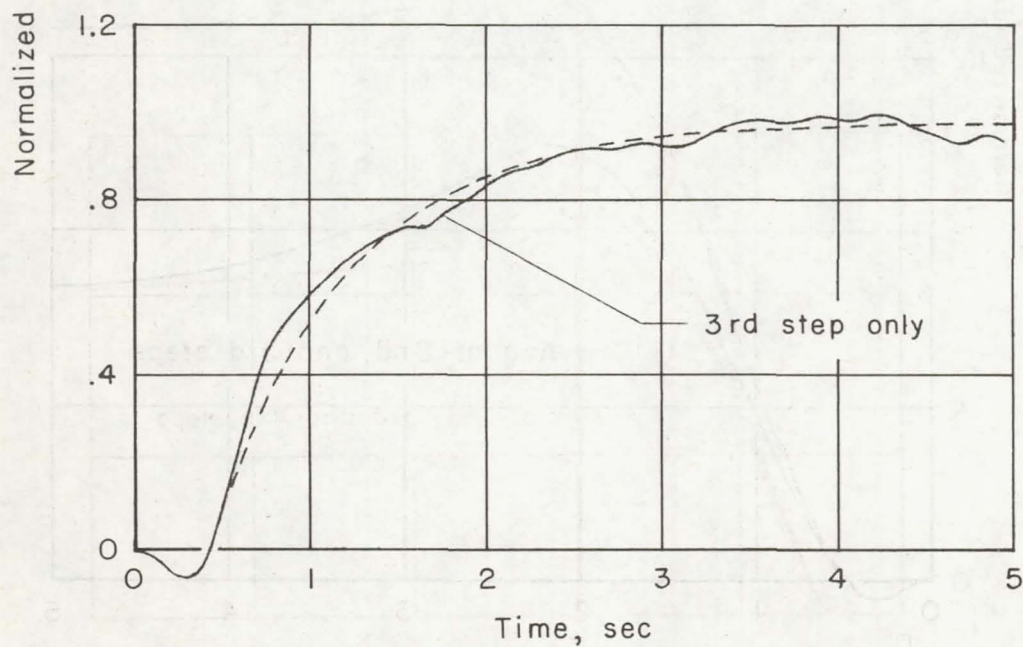
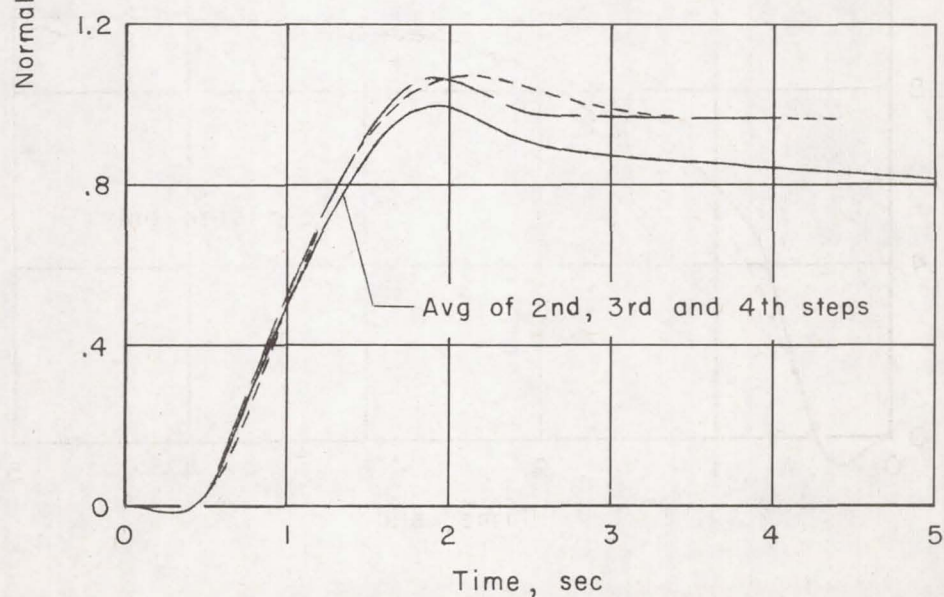
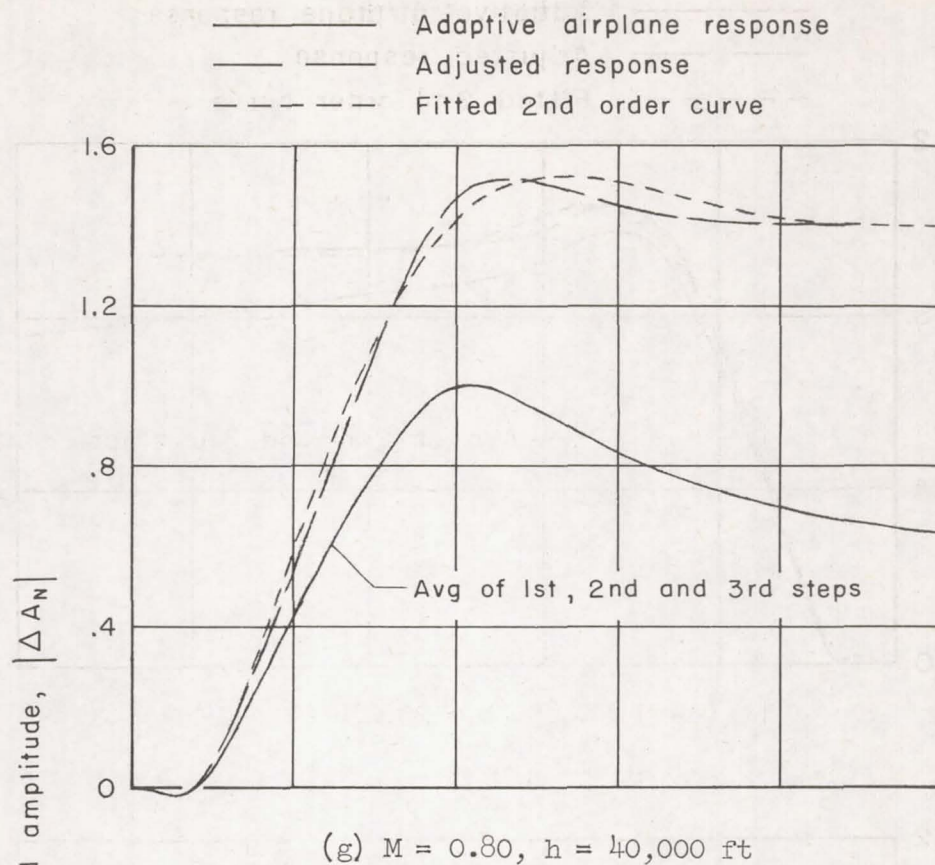
(e) $M = 0.85$, $h = 25,000$ ft(f) $M = 1.15$, $h = 30,000$ ft

Figure 14.- Continued.



(h) $M = 0.96$, $h = 40,000$ ft

Figure 14.- Concluded.